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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 2351

AN EXPERIMENTAL INVESTIGATION OF THE EFFECT OF SURFACE HEATING ON BOUNDARY-LAYER TRANSITION ON A FLAT PLATE IN SUPERSONIC FLOW

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AN EXPERIMENTAL INVESTIGATION OF THE EFFECT OF SURFACE HEATING ON BOUNDARY-LAYER TRANSITION ON A FLAT PLATE IN SUPERSONIC FLOW

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SUMMARY

Wind-tunnel tests have been performed to investigate the effect of surface heating on boundary-layer transition on a flat plate. The tests were performed at a nominal Mach number of 2.40 and a free-stream temperature of -205° F, and the data were obtained at nominal plate temperature levels of 60° (adiabatic recovery temperature), 100°, 140°, 180°, and 260° F over a length Reynolds number range from 0.475×10^{6} to 3.93×10^{6}.

The identification of the onset and end of transition was made by inspection of the curves of surface-tube Mach number reading as a function of length Reynolds number obtained through the transition region. Boundary-layer-velocity profiles were obtained at points corresponding to the onset and end of transition to enable the computation of critical Reynolds numbers based on boundary-layer dimensions.

The transition Reynolds numbers based on the length of run from the plate leading edge, displacement, momentum, and boundary-layer thickness with fluid properties defined alternatively at the free-stream and wall temperatures are presented in graphical form. The results obtained indicate that an increase in surface temperature has a marked influence on decreasing the Reynolds number of transition, although the change in this Reynolds number per unit change in temperature decreases with increased surface temperature.

Values of the average skin-friction coefficient of the laminar boundary layer which was present over the leading 6 inches of the plate were calculated from the results of the boundary-layer surveys and compared with theory. The rate of change of the average skin-friction coefficient with length Reynolds number agrees with theory, although the absolute values are about 35 percent higher than theoretical values obtained by Crocco-Conforto. The present experimental data are in excellent agreement with other experimental results.

INTRODUCTION

Theoretical analyses have shown that there is an influence of surface temperature on the stability of the laminar boundary layer. The addition of heat to a gas from a solid surface has been shown to have a destabilizing effect on the laminar layer through the action of buoyant forces due to the density gradient in the fluid above the surface. It may be shown from a consideration of the equation of motion that such heat addition to a gas will produce an inflected velocity profile; this type of profile has been shown (reference 1) to be more unstable at high Reynolds numbers than a regular-type profile which is convex throughout. Lees (reference 2) has demonstrated that the minimum critical Reynolds number at which self-excited disturbances (slowly growing oscillations, not turbulence) first appear in the boundary layer is not only decreased with heat addition, but that the maximum rate of amplification of the self-excited disturbances propagated along the surface is inversely proportional to approximately the square root of the minimum critical Reynolds number. This would mean, for given external conditions, that (insofar as this mechanism is the source of transition) the length interval between the first appearance of self-excited disturbances and the onset of transition should be shorter for a lower critical Reynolds number. Thus, from theoretical considerations of laminar boundary-layer stability, it may be concluded that transition is advanced by the addition of heat to the fluid as compared with the adiabatic case at the same Mach number.

Liepmann and Fila have shown experimentally in reference 3, for low subsonic free-stream velocities, that transition is advanced as a result of heating a flat plate. Frick and McCullough (reference 4) have noted the change of transition point due to heating the upper surface of an NACA 65,2-016 airfoil at three different chordwise locations: (1) ahead of the minimum pressure point, (2) along the entire laminar run, and (3) at the nose. Their results indicate a decrease in the Reynolds number of transition due to heat addition. The magnitude of the effect on transition is dependent upon the region of application of the heat. Scherrer (references 5 and 6) has shown that transition is advanced by adding heat and delayed by withdrawing heat from a gas flowing supersonically over a 20° cone.

Since the available information on the effect of surface heating on boundary-layer transition is limited in scope, it was felt that additional quantitative experimental data, especially for flat plates in supersonic flow, would be desirable. The present experimental program was initiated to study the movement of the transition point on a flat plate in supersonic flow (Mach number = 2.40) for five nominal surface temperature levels, 60° (adiabatic recovery temperature), 100°, 140°, 180°, and 260° F; to examine the characteristics of the boundary layer immediately preceding and following transition; and, finally, to correlate the information in a usable manner. A further result of the

experimental work was the determination of the average skin-friction coefficient of the laminar boundary layer and its comparison with available theory.

SYMBOLS

C_f average skin-friction coefficient

M Mach number

p static pressure

q dynamic pressure

T absolute temperature

U velocity

x distance from leading edge of plate

y distance normal to plate

γ ratio of specific heats

δ boundary-layer thickness

δ^* boundary-layer displacement thickness

θ boundary-layer momentum thickness

ν kinematic viscosity

ρ density

Subscripts

w wall conditions

∞ free-stream conditions

DESCRIPTION OF EQUIPMENT

Wind Tunnel

The experimental investigation was made with a heated flat plate model in the Ames 6-inch heat-transfer tunnel. The tunnel and auxiliary equipment are described in detail in reference 7.

Flat-Plate Model

The test model, shown schematically in figure 1, was constructed from stainless steel. The model was 16 inches long, 5-1/2 inches wide, and 1/2 inch thick. The upstream end of the plate was chamfered to a 15° angle, and the leading edge was rounded to an approximate radius of 0.003 inch to avoid feathering. A 3-inch-wide by 3/8-inch-deep groove was milled in the bottom of the plate along the center line to permit installation of the electric heater units and plate temperature thermocouples. Two similar grooves 1/2 inch wide located in the bottom of the plate near the sides provided access to the static-pressure orifices. A 1/16-inch-thick cover plate on the bottom sealed these grooves and formed an air space providing insulation between the top and bottom surfaces.

The thermocouples were made from calibrated iron and constantan wires with each wire peened separately 1/4 inch apart spanwise into the underside of the top surface of the plate. The plate thermocouples indicate temperatures 1/16 inch below the upper plate surface at 1/2-inch intervals along the center line.

The electric heaters were made of nichrome wire set in wire-size grooves milled into the top side of thin transite sections measuring 3 inches long by 1/2 inch wide. The heaters were set into the main center groove at 1/2-inch intervals along the center line with the first positioned at $x = 1.2$ inches. A thin sheet of mica, 0.005 inch thick, insulated the heater elements from the upper steel surface.

The static-pressure orifices, 0.0135 inch in diameter, were spaced 1 inch apart chordwise, alternately on two lines, each line located 1 inch from the corresponding side of the plate. This arrangement allowed static-pressure readings to be made at 1-inch intervals along the plate.

The test plate was dowelled to the test-section walls to reduce bending and vibration to a minimum. Thin, soft fabric strips provided bearing surfaces between the glass windows and the plate sides and eliminated flow around the sides. The top surface of the test plate was ground and polished to a high finish.

An impact-pressure survey apparatus was mounted above and downstream of the test plate so that impact-pressure surveys could be made completely through the flow boundary layer at the desired test position. The impact-pressure tube, constructed of flattened hypodermic tubing, was rectangular faced, measuring 0.008 inch in height and 0.080 inch in width, with an opening measuring 0.004 inch by 0.075 inch (fig. 1).

TEST PROCEDURE

In order to determine the onset and end of transition, impact-pressure probe readings were made at a fixed position adjacent to the surface of the plate. The Reynolds number was varied by raising or lowering the tunnel stagnation pressure. A change in the type of boundary layer present on the plate was indicated by a marked change in the Mach number as determined from the magnitude of the pressures indicated by the surface impact-pressure probe and an adjacent static-pressure orifice. Although the absolute magnitude of the surface Mach number is not significant as such, its variation for a fixed free-stream Mach number does permit recognition of the three types of boundary layer, namely, laminar, transitional, and turbulent.

The detailed test procedure necessary to obtain curves of surface-tube Mach number as a function of Reynolds number shown in figure 2 was as follows: After the tunnel operating conditions of total pressure and temperature level had been established, the impact-pressure tube was lowered to the plate surface 6 inches from the leading edge for all tests. The plate temperature was adjusted to the desired level by means of 19 rheostat-controlled heaters which enabled the plate temperature to be maintained constant from $x = 1.2$ inches to a position approximately 4 inches downstream of the probe face. Temperature readings were obtained with a rapid-reading recording potentiometer and heater currents were varied until a constant steady-state plate temperature was realized. The final plate-temperature thermocouple voltages were recorded with a manual-balancing potentiometer in order that better accuracy of temperature measurement would be insured. Stagnation-temperature thermocouple voltages were recorded at this time.

In conjunction with the temperature readings, impact- and static-pressure measurements were made at the impact-pressure probe position on the plate surface. The probe was then raised to a position outside the boundary layer, and the static-pressure distribution along the plate and the free-stream impact pressure corresponding to the probe position were measured. The measurements were made on mercury and dibutyl-phthalate manometers with a high vacuum used as a common reference for all manometers. In order to keep the pressure-measuring errors to a minimum, the low impact-pressure and all static-pressure values were read with dibutyl-phthalate manometers.

At the completion of the preliminary work necessary to establish the curves shown in figure 2, the Reynolds numbers denoting the onset and end of transition were determined by inspection of the curves, arbitrarily assigning the beginning of transition to the point on each curve where the surface Mach number started to rise from its minimum value. Correspondingly, the end of transition was chosen as the position on each curve where the Mach number started to decrease appreciably from a straight line passing through the maximum Mach number values. Because of the change in shape of the curve in the region judged to be the end of transition for a plate temperature of 260° F, a boundary-layer profile at the selected point of the end of transition was obtained and compared with a fully developed turbulent boundary-layer profile obtained at a higher Reynolds number and the same plate temperature. The two profiles were in excellent agreement, indicating that fully developed turbulent flow existed at the selected point in question.

For each end point of the transition regions, the total pressure level which gave the prescribed Reynolds number was set and the plate surface temperature was adjusted to the desired level. A complete boundary-layer survey was then made, starting at the plate surface and traversing to the free stream. The height of the impact-pressure tube above the plate surface was measured with a dial indicator mounted on the vertical post of a cathetometer. The least count of the indicator was 0.0001 inch. The telescope of the cathetometer was sighted through one test-section window on a fine line scribed on the impact-pressure probe stiffener. This line was parallel to the plate surface and scribed sufficiently high above the lower surface of the tube to be outside the boundary layer, thereby eliminating errors in tube height due to refraction effects. It is believed that the tube position could be measured to ±0.001 inch.

The time lag to obtain an impact-pressure measurement varied with the absolute pressure measured and was in the order of 10 to 30 minutes. A pressure time history was made for each impact-pressure reading during the surveys to establish the steady-state value.

DATA REDUCTION

All boundary-layer impact-pressure data were first reduced in terms of Mach number. Faired curves (figs. 3 and 4) were drawn for the boundary-layer profiles from which values of Mach number and ordinate stations were taken. Local temperature and velocity distributions were then evaluated employing a relation of Crocco's (reference 8), which assumes a Prandtl number of 1, and the fact that temperature may be

expressed as a function of velocity and the measured Mach number. Crocco's equation may be expressed as:

$$\frac{T}{T_\infty} = \frac{T_W}{T_\infty} + \left(1 + \frac{\gamma-1}{2} M_\infty^2 - \frac{T_W}{T_\infty}\right)\frac{U}{U_\infty} - \frac{\gamma-1}{2} M_\infty^2 \left(\frac{U}{U_\infty}\right)^2 \tag{1}$$

The laminar and turbulent boundary-layer velocity profiles, at the beginning and end of transition, are plotted in figures 5 and 6, respectively. The temperature distributions along the plate at the various levels at which the surveys were taken are shown plotted in figure 7.

Evaluation of the boundary-layer displacement thickness

$$\delta^* = \int_0^\delta \left(1 - \frac{\rho U}{\rho_\infty U_\infty}\right) dy \tag{2}$$

and the momentum thickness

$$\theta = \int_0^\delta \frac{\rho U}{\rho_\infty U_\infty} \left(1 - \frac{U}{U_\infty}\right) dy \tag{3}$$

was made by numerical integration of the respective functions of density and velocity.

The values of average laminar boundary-layer skin-friction coefficient from the plate leading edge to $x = 6$ inches were evaluated from the equation defining the momentum decrement

$$C_f = \frac{2\theta}{x} \tag{4}$$

RESULTS AND DISCUSSION

The position of the transition point along the plate surface may be influenced both by factors within the boundary layer and by conditions outside the boundary layer. Briefly, the external conditions which affect boundary-layer transition are: pressure gradient, turbulence level, external pressure fluctuations, surface roughness, and the transport process of an external disturbance through a normally laminar boundary layer, termed "transverse contamination" by Charters (reference 9).

In the series of experimental tests described herein, emphasis has been centered on the influence of surface temperature upon the factors within the boundary layer which control the transition point, while an attempt has been made to minimize the effect of external influences on boundary-layer transition. The constancy of the test-section static-pressure is evidenced in figure 8. The magnitude of the tunnel turbulence level is unknown but is believed to be low because of the high contraction ratio and effective damping by six fine-mesh wire screens mounted upstream of the test section. The magnitude of the pressure fluctuations within the test section are believed to be damped by the large receiving chamber upstream of the test section. The measured roughness of the plate surface was found not to exceed a maximum deviation of 25 microinches from the mean profile. The effect of transverse contamination from the tunnel walls has been found to lie beyond the testing region, from evidence given by a luminous film method of detecting this phenomenon. (Cf. reference 10.)

The transition regions corresponding to the five plate temperature levels are defined in figure 2 by the curves of surface-tube Mach number as a function of Reynolds number. The limits of the three regimes of boundary-layer flow, laminar, transitional, and turbulent, are indicated by the dark symbols on each curve. At adiabatic wall temperature (60° F) the Reynolds number defining the extent of the laminar region is 1.25×10^6 and decreases gradually with increasing plate temperature to a value of 0.6×10^6 for a plate temperature of 260° F. The extent of the Reynolds number range for the transition region decreases from about 2×10^6 to a value of 1×10^6 over the same temperature range of 200° F.

Since in this series of tests the Reynolds number was varied by changing the tunnel stagnation pressure, the results in figure 2 may be affected by a possible variation of turbulence level with pressure. The fact that transition extends over the afore-mentioned Reynolds number range indicates one of two possibilities: Either the transition from laminar to turbulent flow is a gradual process, requiring a definite region through which the laminar flow is destroyed, or transition is a sudden process, occurring over a relatively narrow region, which, in itself, fluctuates back and forth along the plate within the region indicated by the survey apparatus. Dryden (reference 11) has shown in subsonic flow that the transition point, defined by him as the point at which the first bursts of turbulence are indicated by hot wire equipment, is subject to rapid to-and-fro movement along the plate surface.

The effect of surface temperature on transition is depicted in a series of curves of Reynolds numbers, based on length of run, boundary-layer thickness, displacement thickness, and momentum thickness for free-stream properties (figs. 9, 10, 11, and 12) and for wall properties (figs. 13, 14, 15, and 16), as a function of the ratio of wall temperature to free-stream temperature. The general conclusions that may be drawn from an examination of all these curves are as follows: The

Reynolds numbers at the onset and end of transition decrease with increased temperature ratio; however, the Reynolds number at the end of transition decreases more rapidly than that at the onset of transition. Therefore, the extent of the Reynolds number range of transition decreases with increase in temperature ratio. Examination of figures 9 through 16 shows a change in the slope of the curves of Reynolds number as a function of temperature ratio which would indicate a decrease in the effect of surface temperature upon the change in Reynolds number for increasing surface temperature. This effect has been noted by Frick and McCullough in their work on heated low-drag airfoils at subsonic speeds (reference 4).

Transition Reynolds number values, based on boundary-layer momentum thickness, determined by Frick and McCullough on a low-drag airfoil in subsonic flow show a decrease of about 33 percent as the airfoil surface temperature was increased approximately 100° F. The present tests indicate a decrease in the corresponding transition Reynolds number values, shown in figure 12, of about 31 percent (1050 to 720) for a surface temperature increase of 100° F above adiabatic recovery temperature. This agreement may be fortuitous in light of the fact that the tests were performed under such dissimilar conditions.

In analyzing the data obtained in the tests, an attempt was made to evaluate the transition Reynolds numbers with fluid properties based on an intermediate temperature between the free-stream value and that at the solid surface in accordance with suggestions made in a report by Allen and Nitzberg (reference 12). It was found that no particular advantage could be gained by employing fluid properties at any intermediate temperature. The present experimental results do not provide sufficient basis to state that any particular length parameter or that any particular evaluation of fluid properties better correlate the variation of the Reynolds number of transition with surface temperature.

The average skin-friction coefficient for the laminar boundary-layer at each of the plate temperatures is shown in figure 17. The values are compared to the theory of Crocco and Conforto as presented in reference 13. The rate of change of the average skin-friction coefficient with Reynolds number agrees with the theory; however, the absolute values are approximately 35 percent higher than the theoretical values, but are in excellent agreement with the experimental results obtained by Blue (reference 14) on an unheated flat plate in supersonic flow. A further study of the problem would be necessary before an adequate explanation of the discrepancy could be made.

CONCLUDING REMARKS

Surface heating has a marked effect upon the transition Reynolds number. This fact has been demonstrated qualitatively before for both

subsonic and supersonic flow, but adequate quantitative data have not been presented for supersonic flow. The results of the present tests indicate that, for a surface temperature increase of 200° F, the Reynolds number at the onset of transition based on boundary-layer length of run and free-stream fluid properties was decreased from 1.25×10^6 to 0.6×10^6, while corresponding decreases in Reynolds numbers based on displacement, momentum, and boundary-layer thickness at the onset of transition are from 6300 to 4880, from 1050 to 632, and from 13,400 to 8830, respectively. Although the results cannot be construed as an absolute quantitative measure of the surface temperature effect, they provide a general picture which gives added knowledge of the problem.

Although the influence of surface temperature on transition is decidedly marked, the results indicate that the change in Reynolds number of transition per unit change in temperature ratio decreases for increasing temperature ratio. This effect has been shown previously by Frick and McCullough.

Without excluding the possibility that transition may occur in a narrow band which oscillates within the transition region, this transition region has a length of the same order of magnitude as the laminar region. The extent of this transition region decreases with an increase in surface temperature.

Values of the skin-friction coefficient determined experimentally for the laminar boundary layer on a heated plate show excellent agreement with independent results obtained by Blue on an unheated flat plate in supersonic flow, although both sets of experimental data are about 35 percent higher than available theoretical values. The reasons for the discrepancy are not known at this writing.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif., February 27, 1951.

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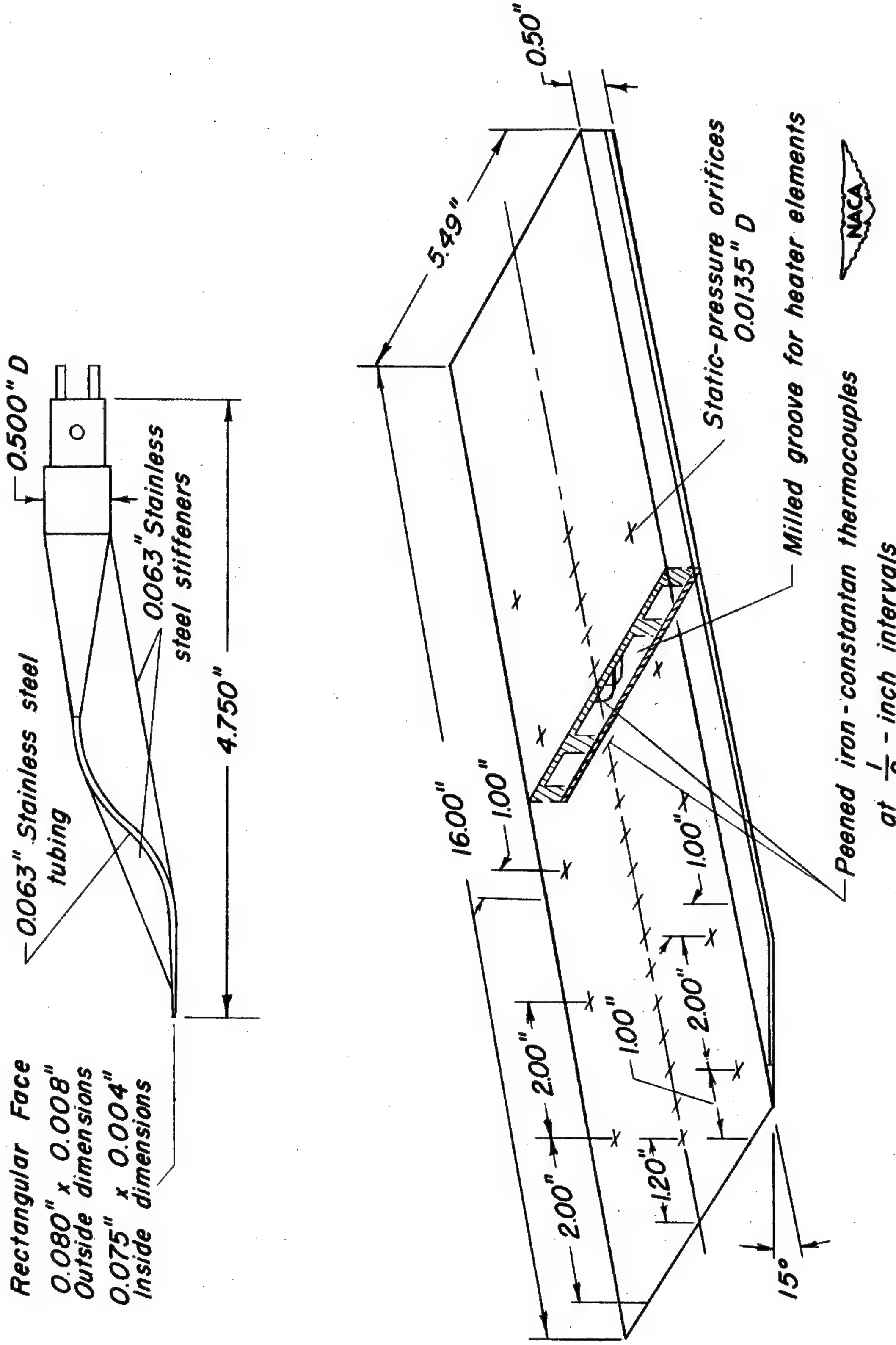

Figure 1.- Sketch of the heated flat-plate model and impact-pressure probe.

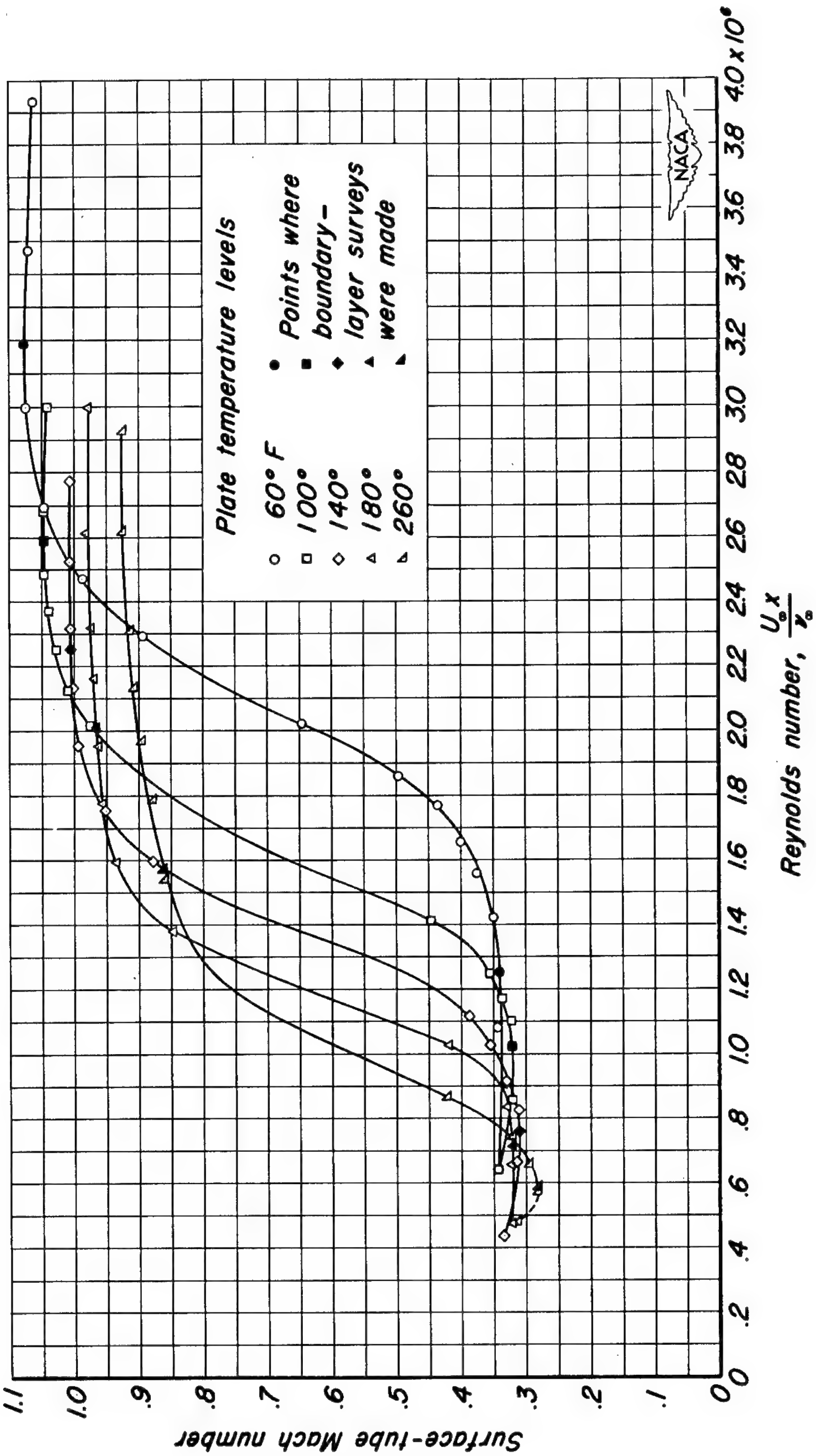

Figure 2.- The variation of surface-tube Mach number with Reynolds number.

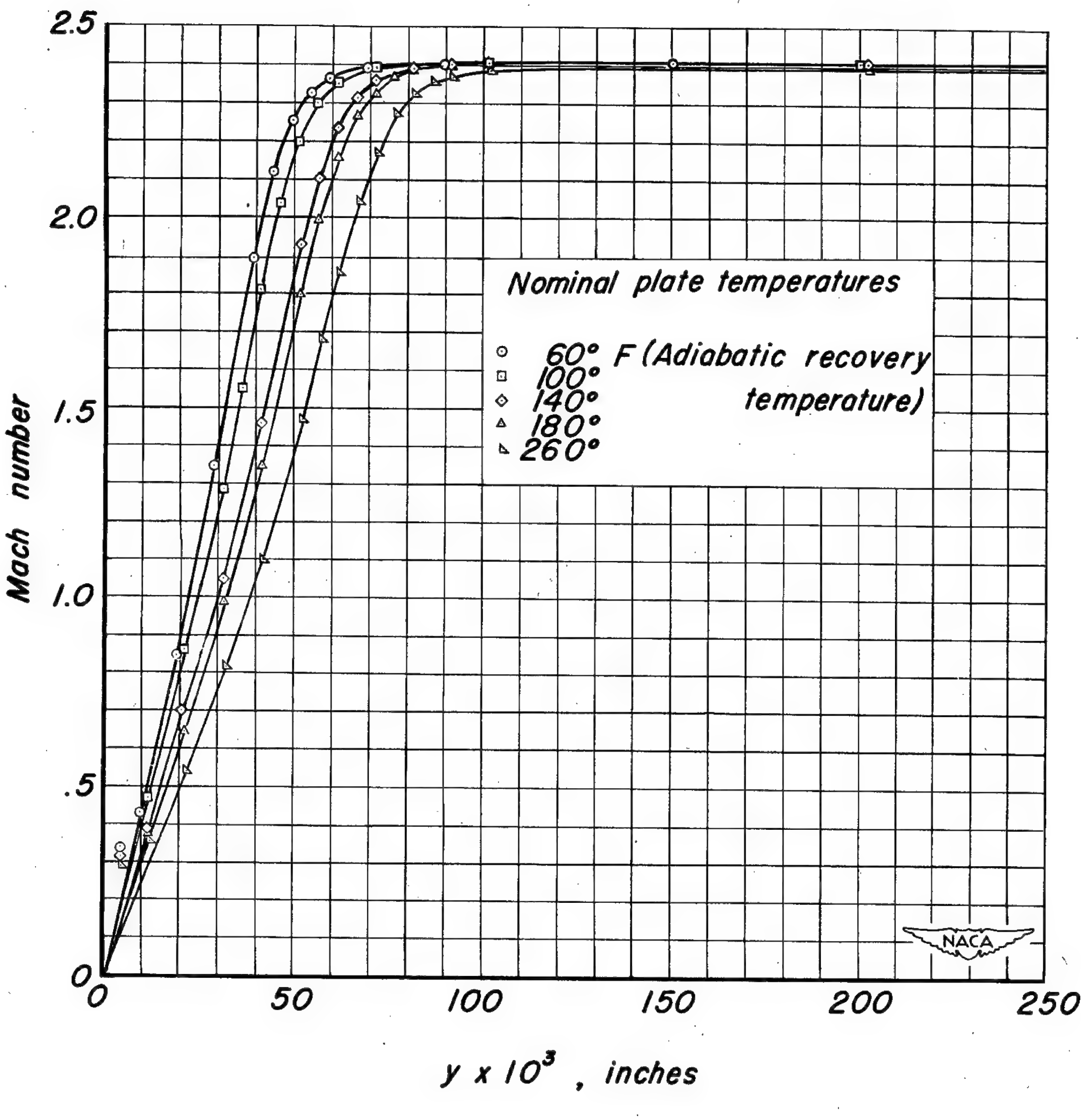

Figure 3.— Laminar boundary-layer Mach number distributions at several nominal plate-temperature levels.

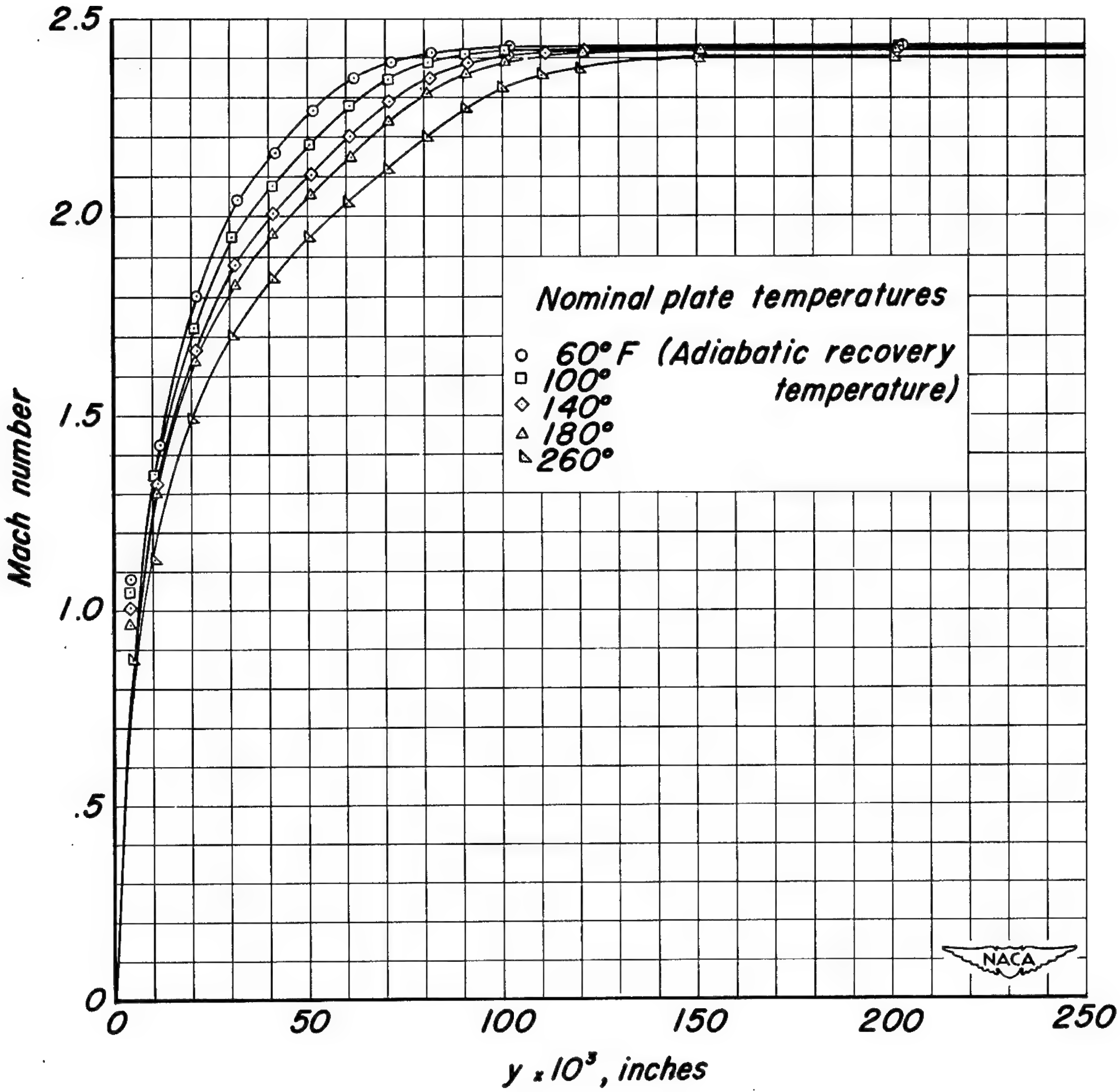

Figure 4.- Turbulent boundary-layer Mach number distributions at several nominal plate-temperature levels.

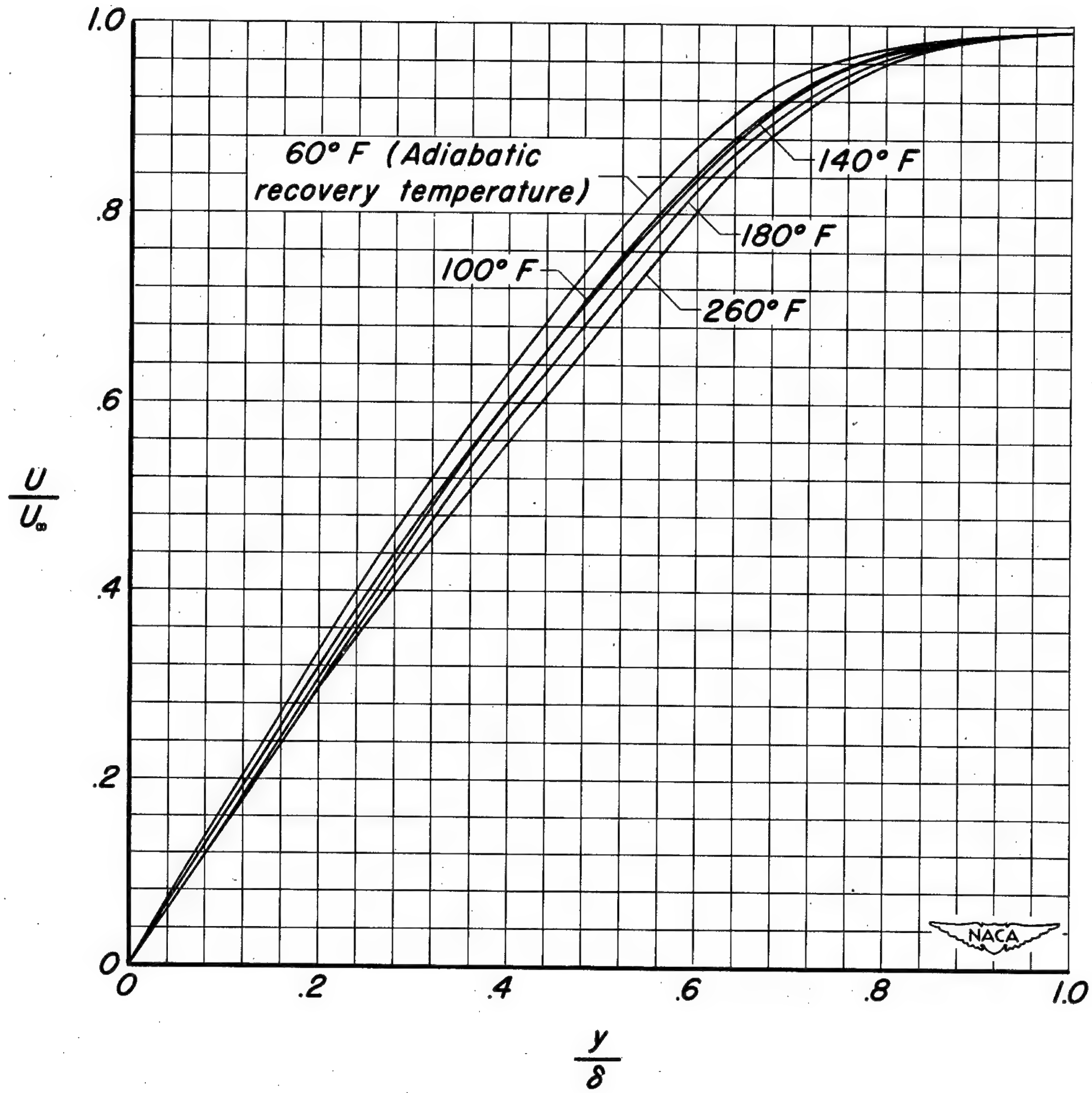

Figure 5.- Laminar boundary-layer velocity distributions at several nominal plate-temperature levels.

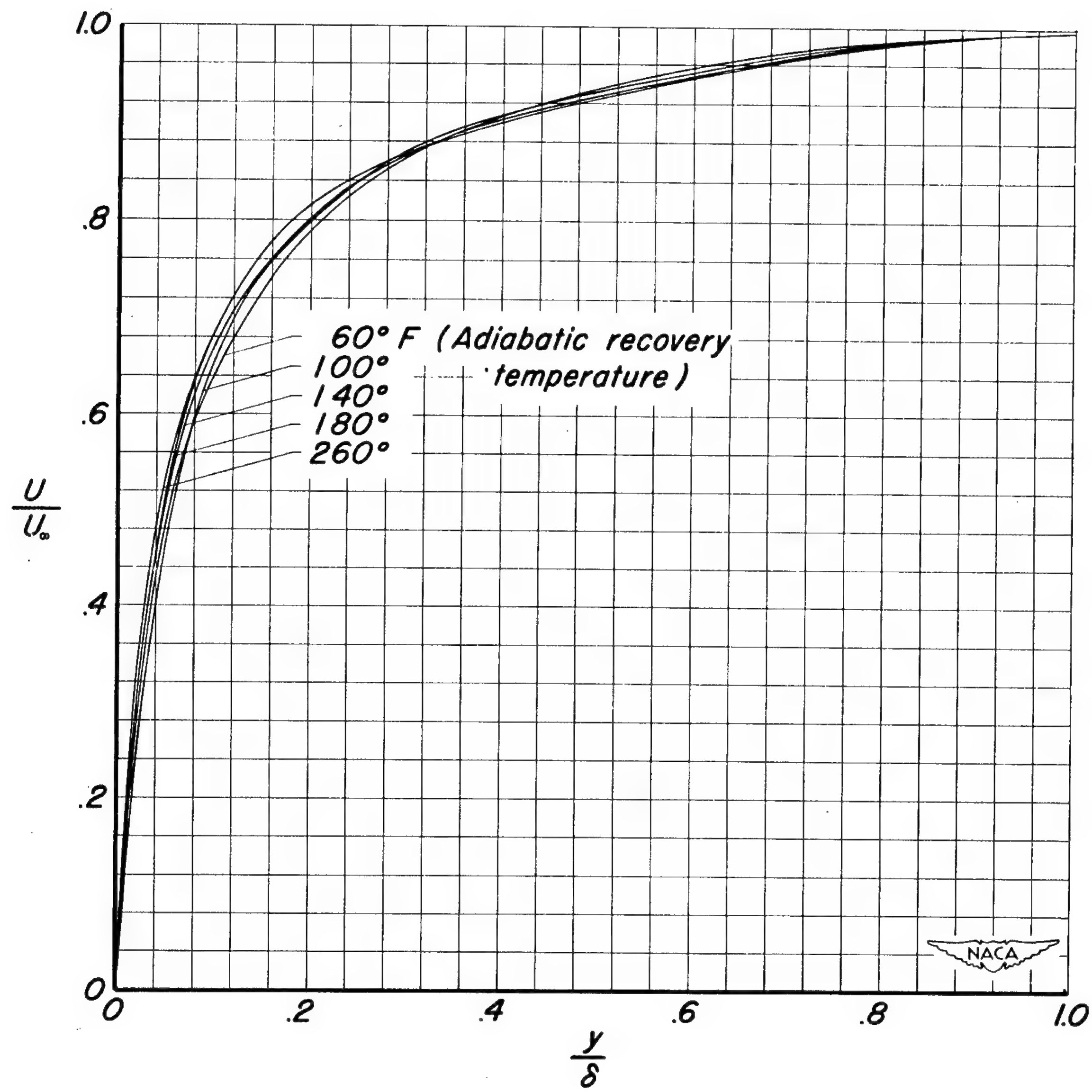

Figure 6.—Turbulent boundary-layer velocity distributions at several nominal plate-temperature levels.

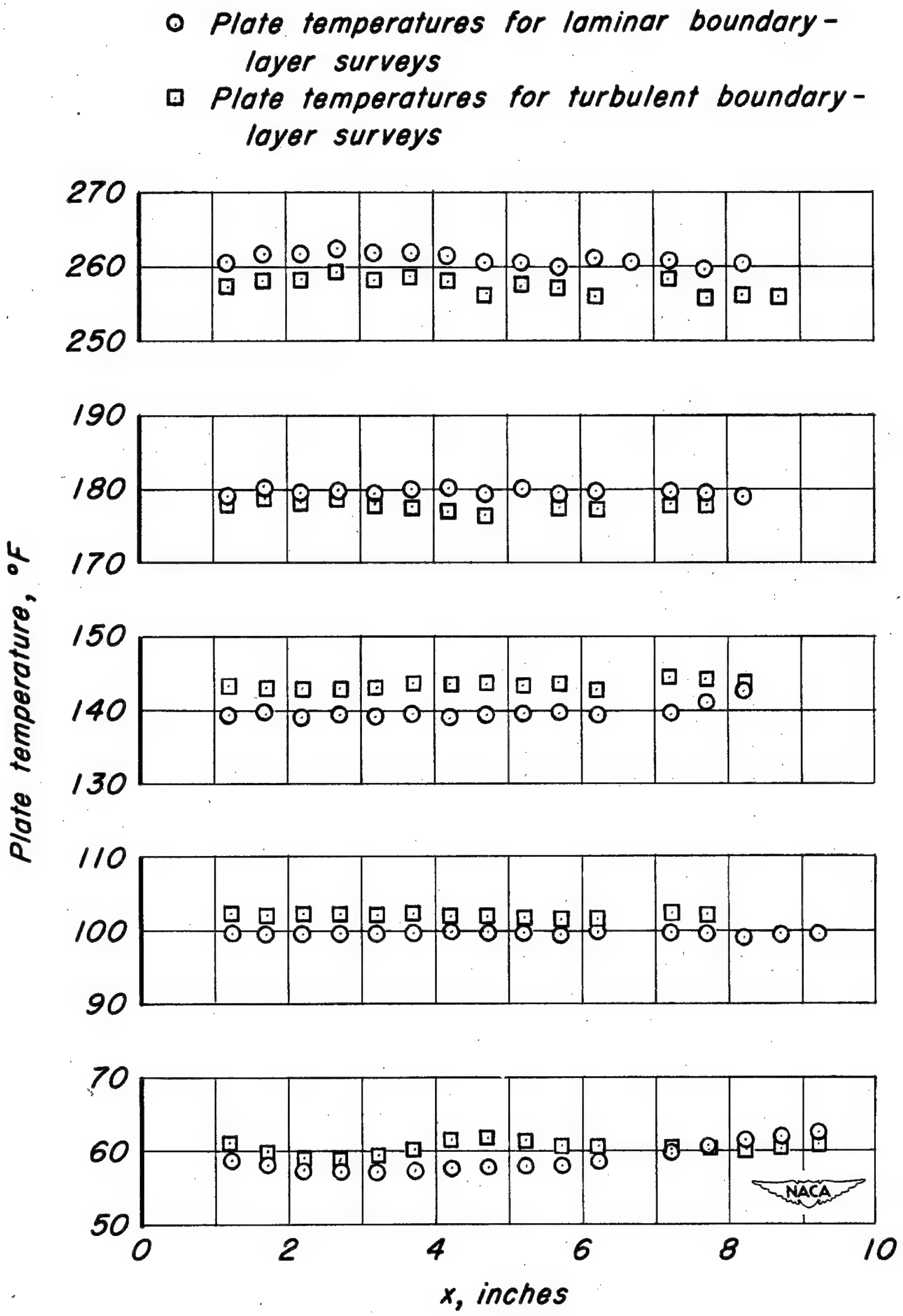

Figure 7.– Representative plate-temperature distributions.

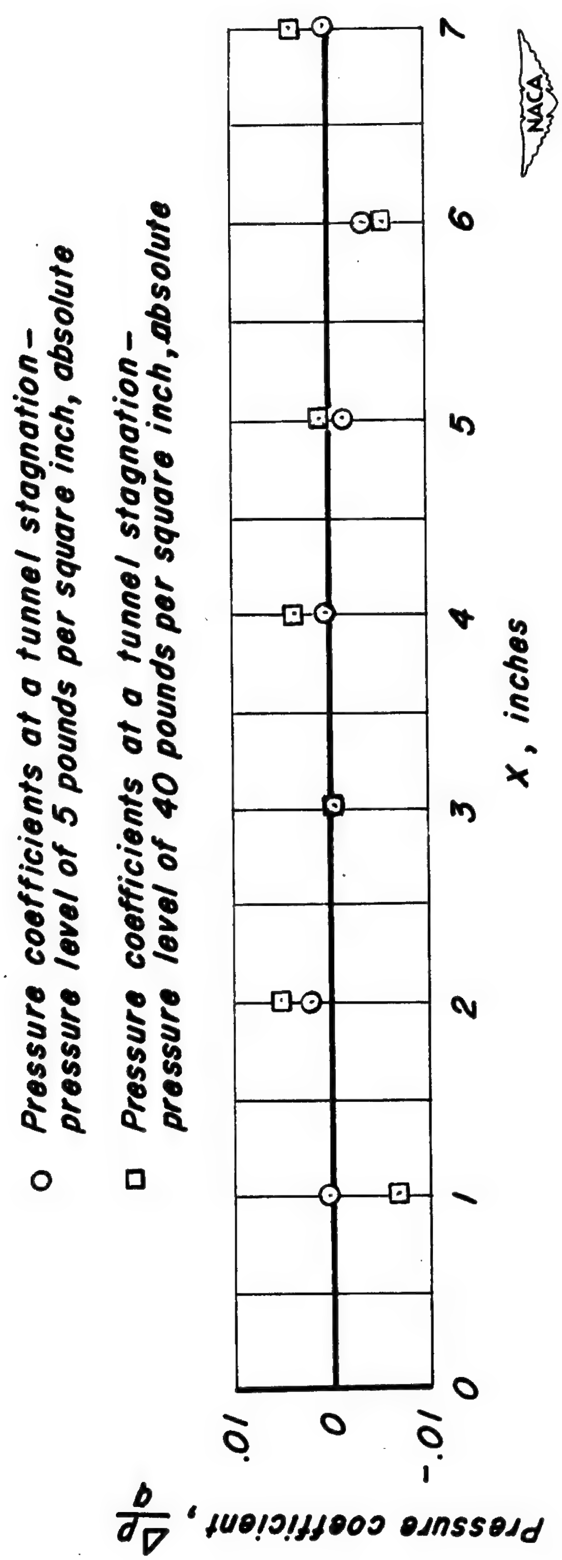

Figure 8. - Static-pressure distribution along the test model.

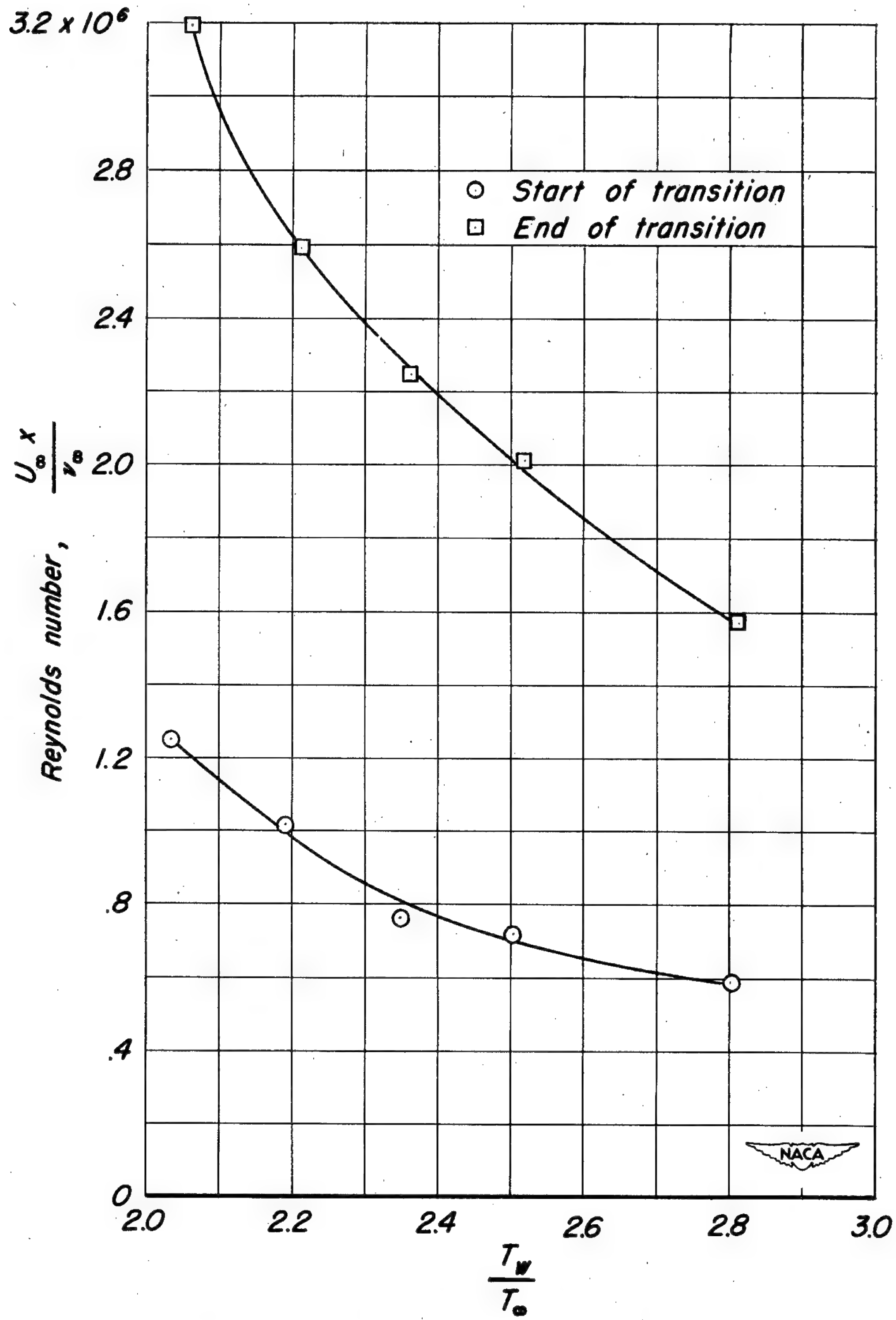

Figure 9.– Transition Reynolds numbers based on x distance and free-stream properties.

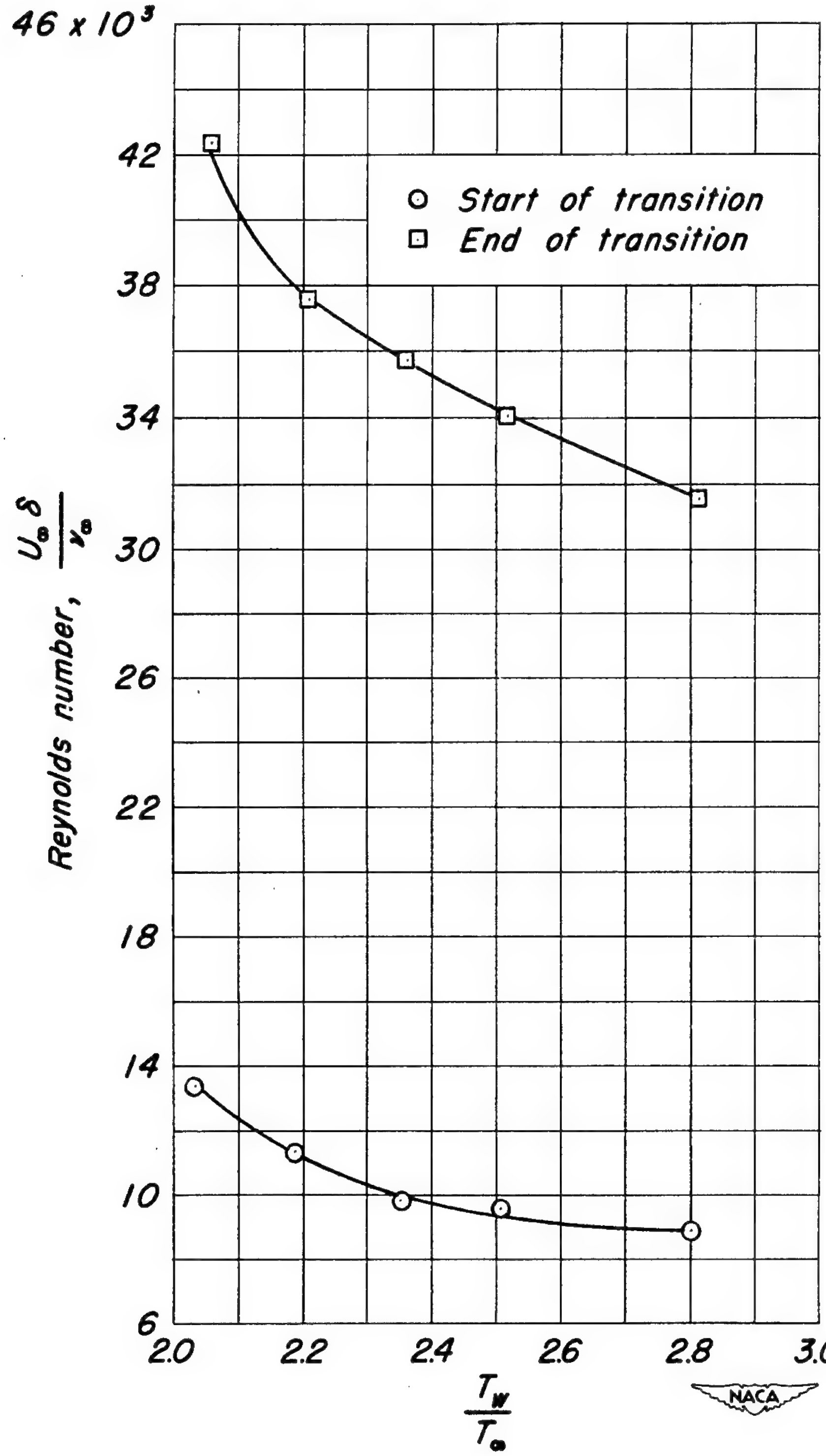

Figure 10.— Transition Reynolds numbers based on boundary-layer thickness corresponding to $U = 0.995 U_\infty$ and free-stream properties.

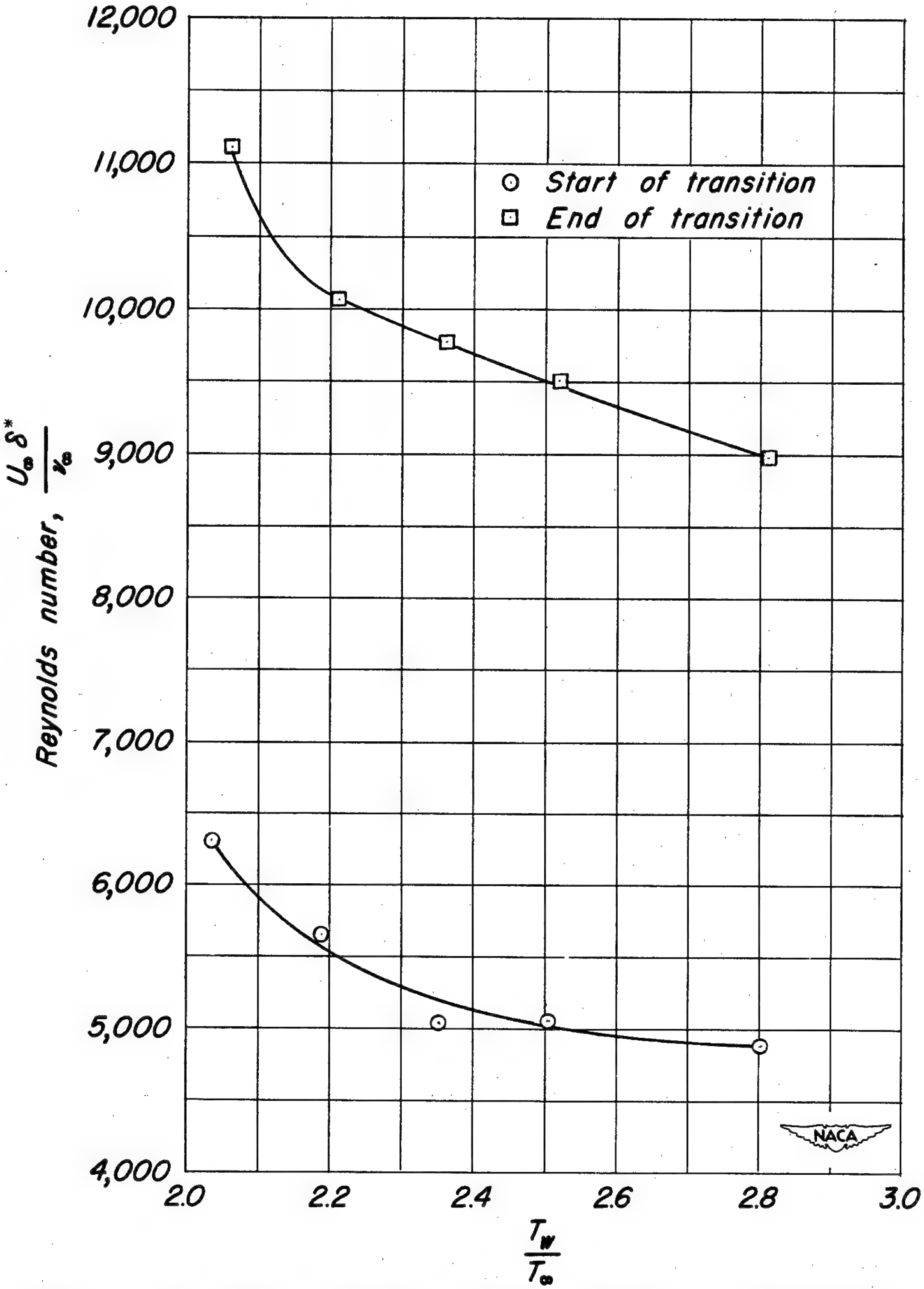

Figure 11.—Transition Reynolds numbers based on displacement thickness and free-stream properties.

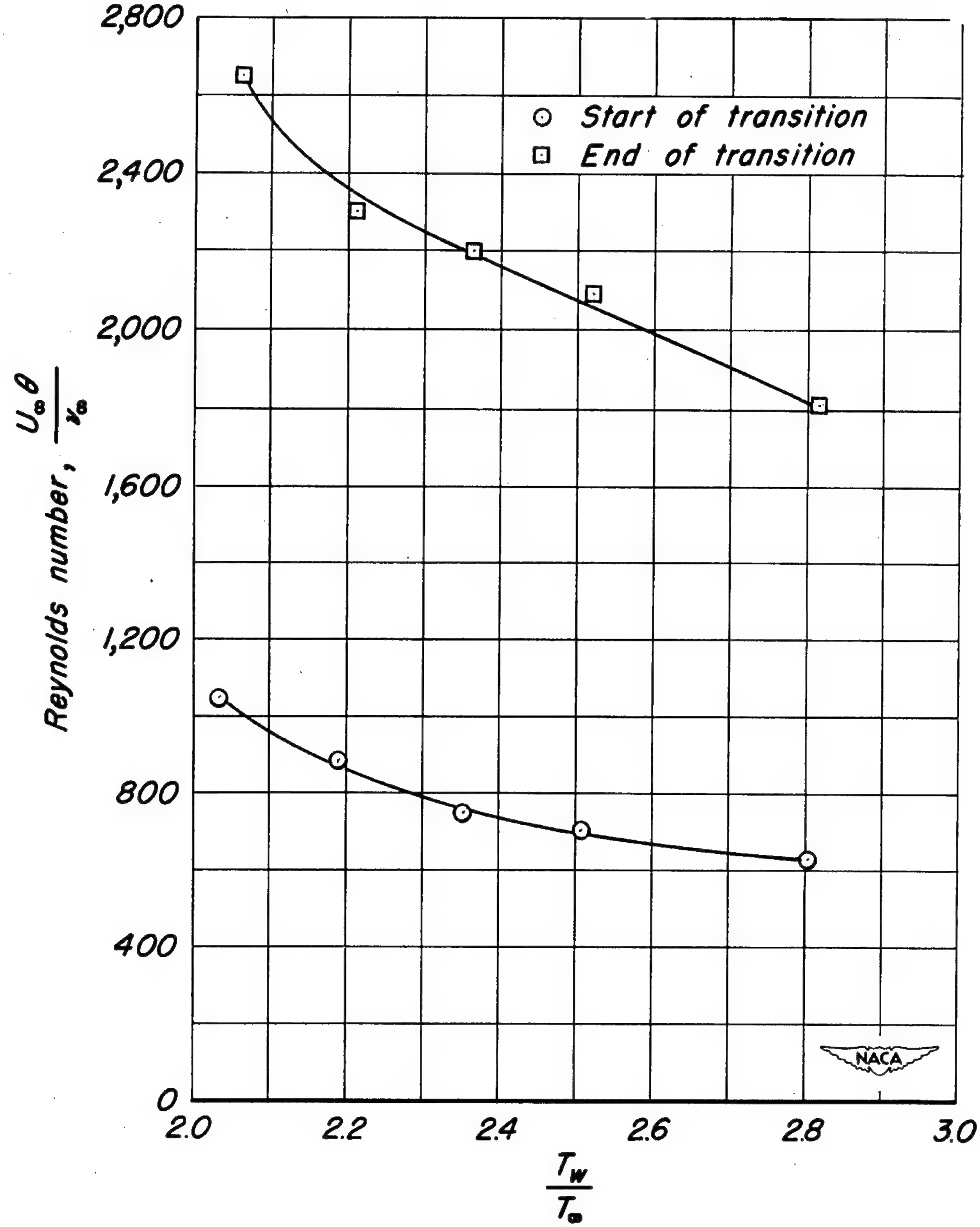

Figure 12.- Transition Reynolds numbers based on momentum thickness and free-stream properties.

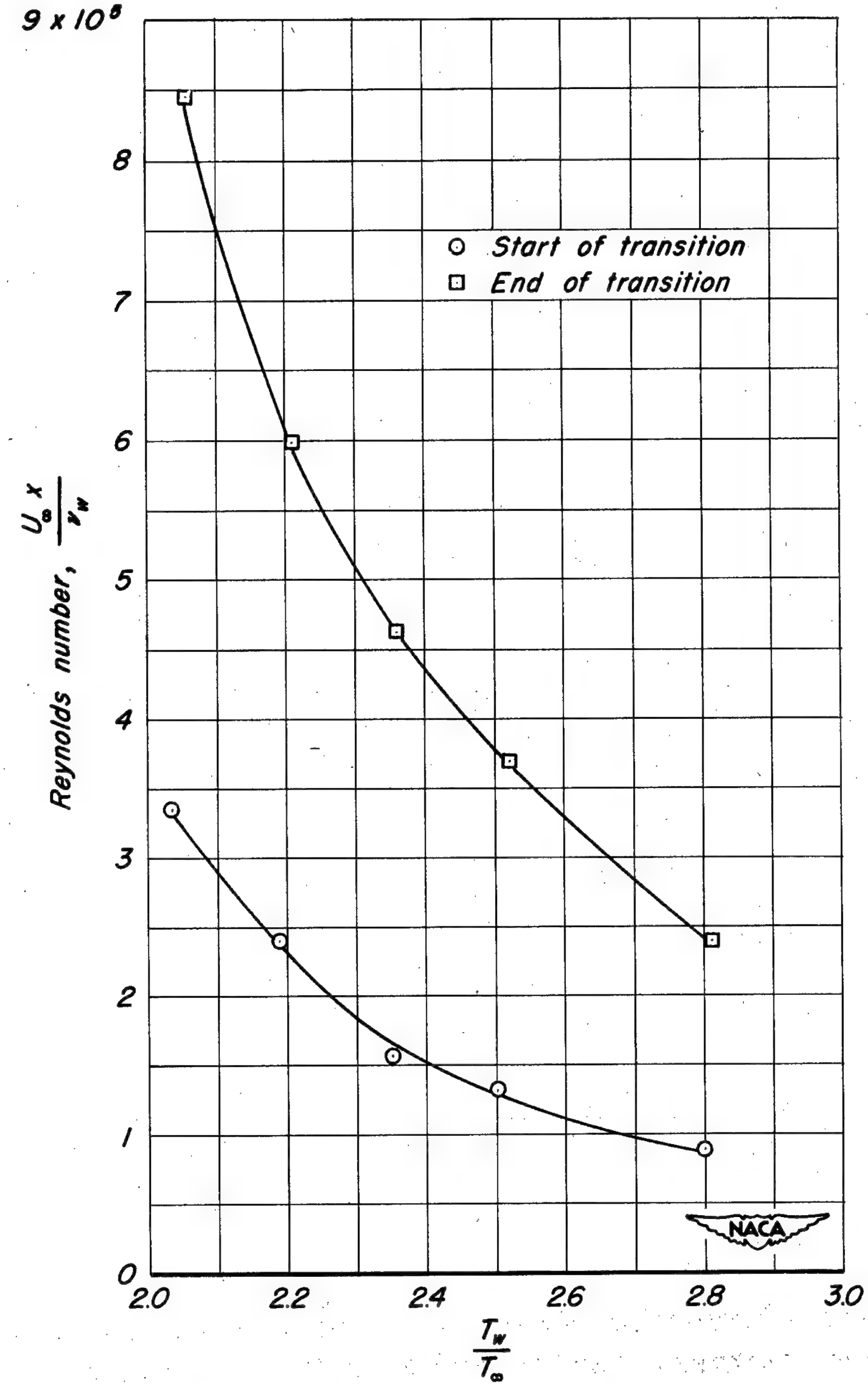

Figure 13.- Transition Reynolds numbers based on x distance and wall properties.

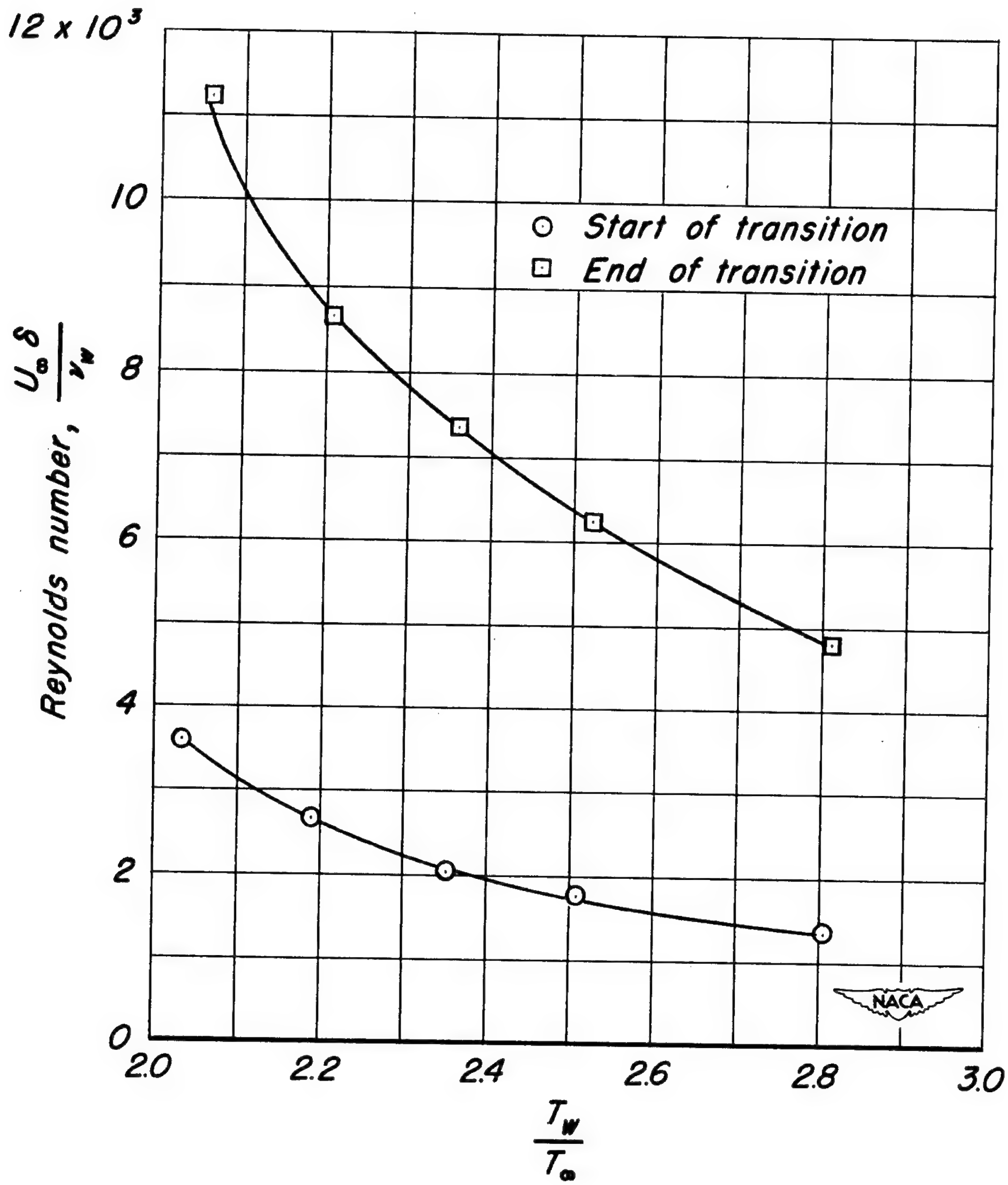

Figure 14.- Transition Reynolds numbers based on boundary-layer thickness corresponding to $U = 0.995\,U_\infty$ and wall properties.

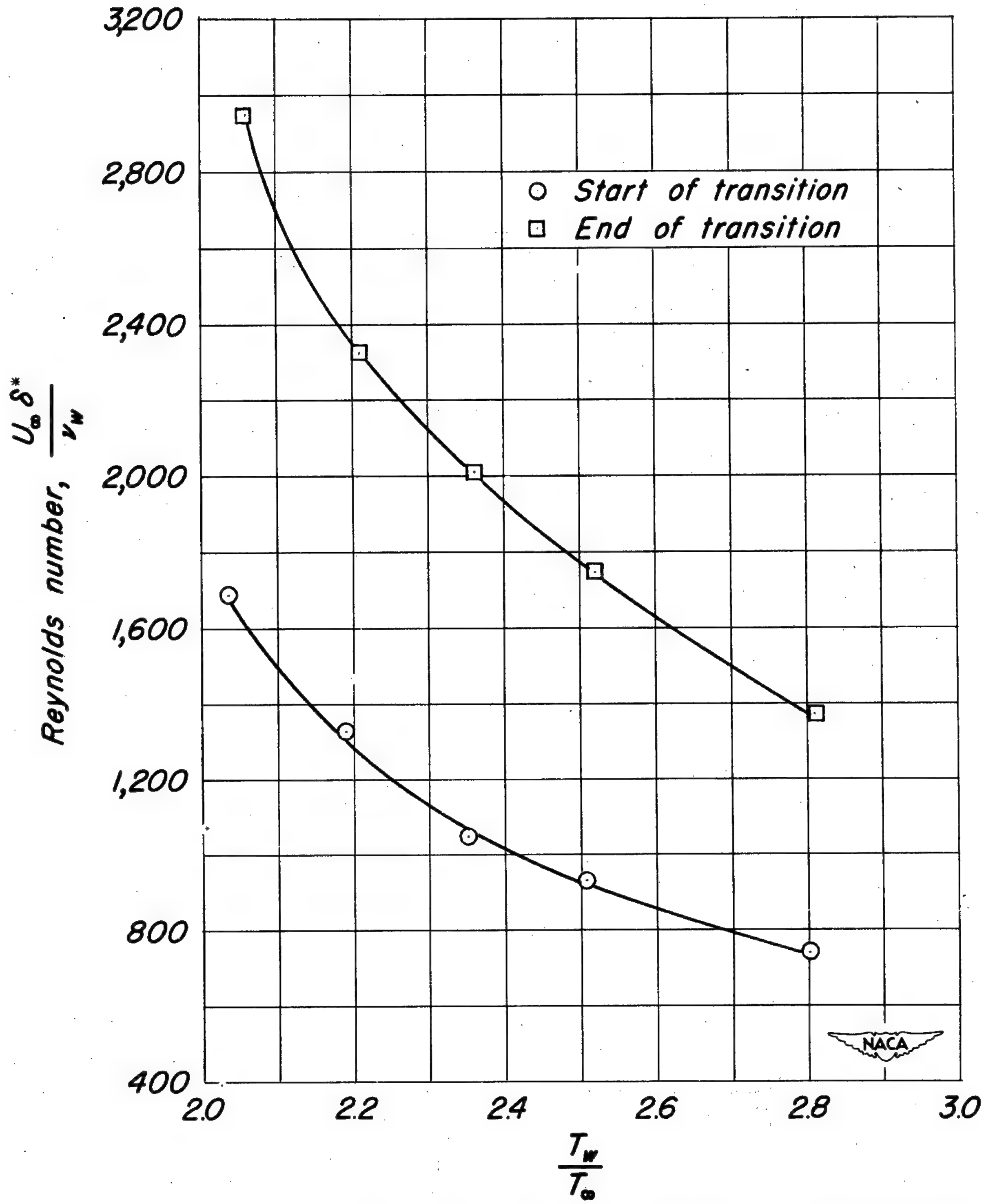

Figure 15.- Transition Reynolds numbers based on displacement thickness and wall properties.

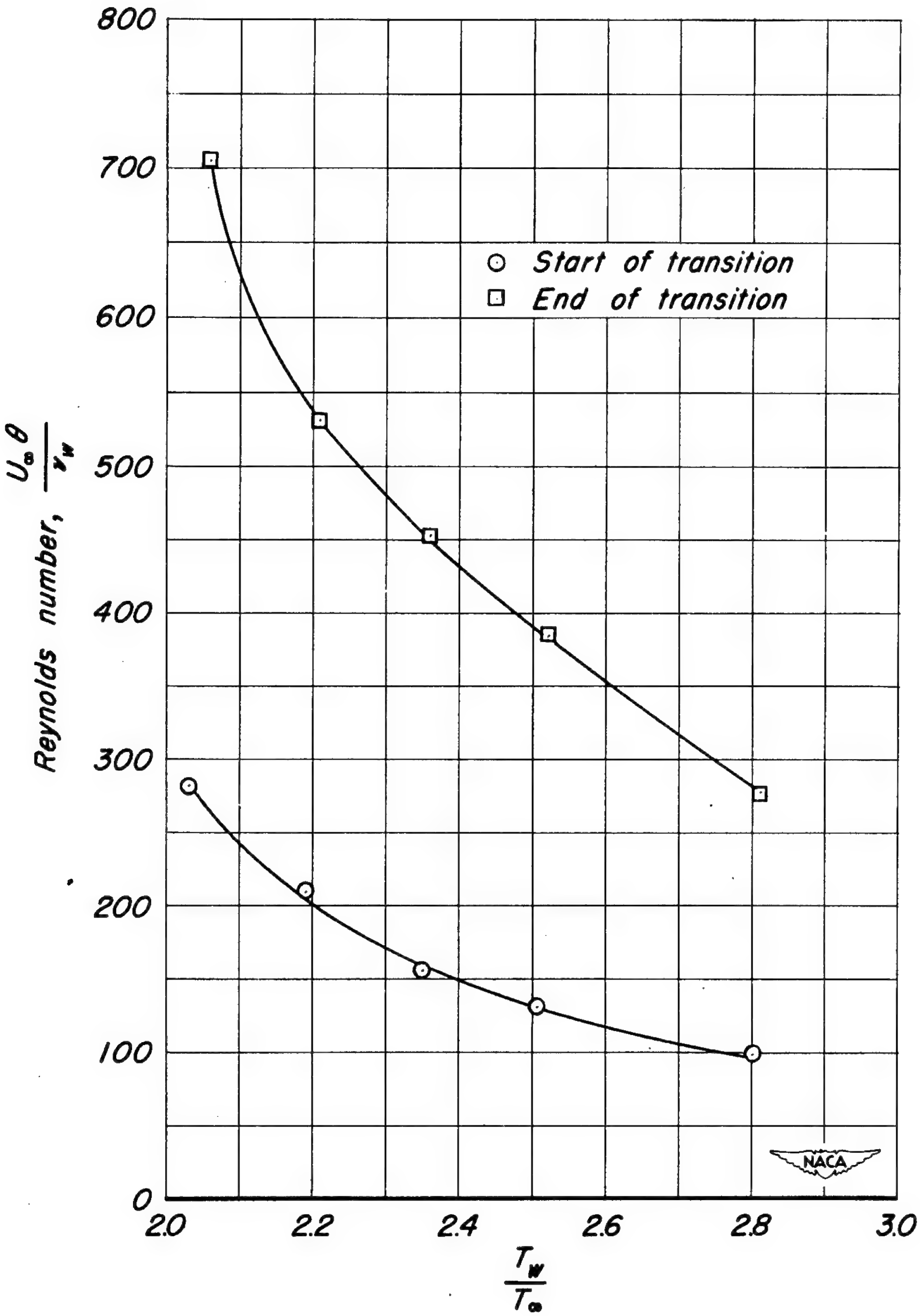

Figure 16.- Transition Reynolds numbers based on momentum thickness and wall properties.

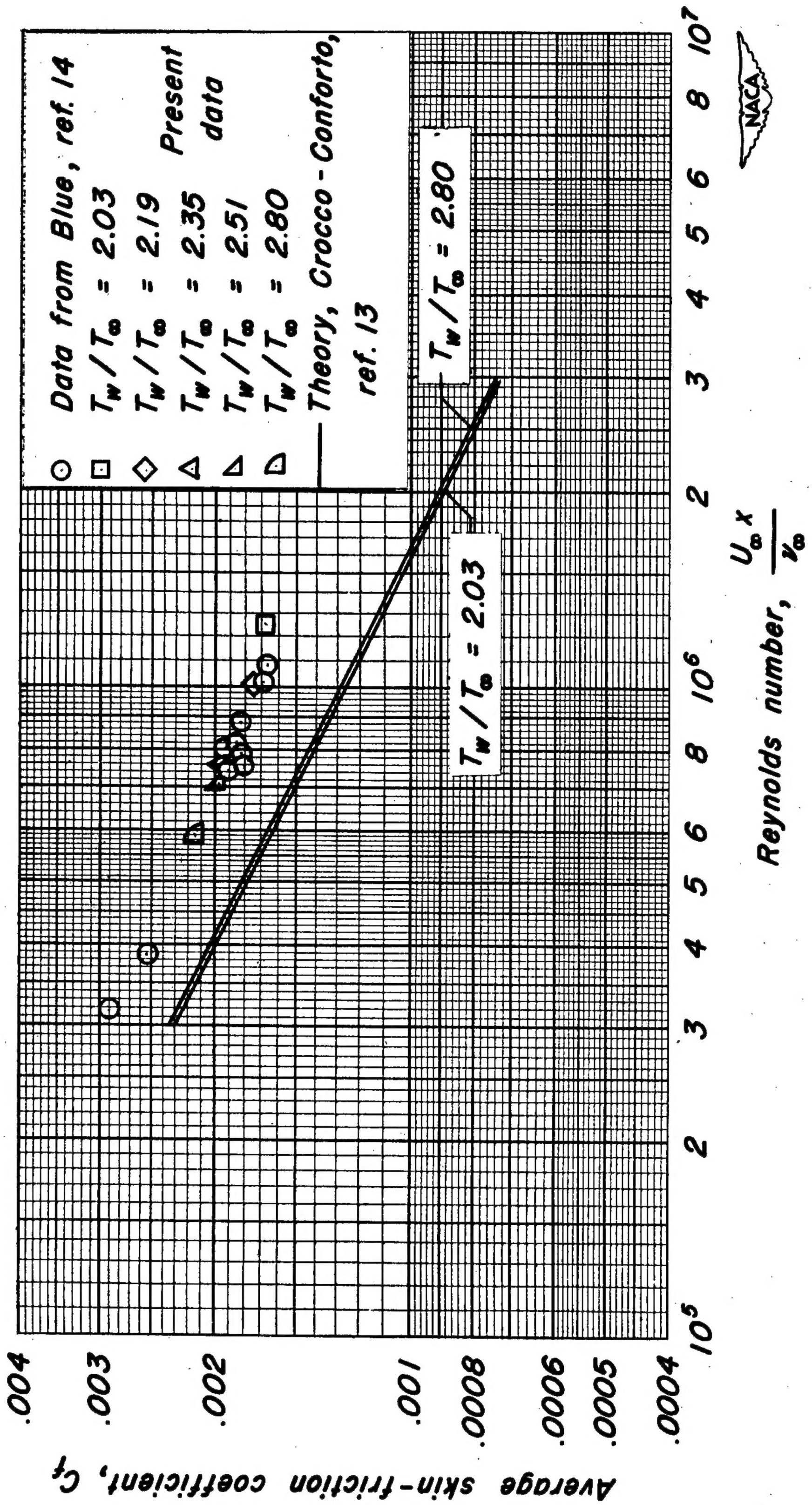

Figure 17.- Average skin-friction coefficient of the laminar boundary layer on a flat plate.

Flow, Laminar 1.1.3.1

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Flow, Turbulent 1.1.3.2

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Abstract

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Abstract

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